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Pulse Detonation Engines for High Speed Flight

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PULSE DETONATION ENGINES FOR HIGH SPEED FLIGHT

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Introduction

Revolutionary concepts in propulsion are required in order to achieve high-speed cruise capability in the atmosphere and for low cost reliable systems for earth to orbit missions. One of the advanced concepts under study is the air-breathing pulse detonation engine.[1] Additional work remains in order to establish the role and performance of a PDE in flight applications, either as a stand- alone device or as part of a combined cycle system. In this paper, we shall offer a few remarks on some of these remaining issues, i.e., combined cycle systems, nozzles and exhaust systems and thrust per unit frontal area limitations. Currently, an intensive experimental and numerical effort is underway in order to quantify the propulsion performance characteristics of this device. ***In this paper we shall highlight our recent efforts to elucidate the propulsion potential of pulse detonation engines and their possible application to high-speed or hypersonic systems.***

Analyses

Cycle Analysis

In this paper, we begin with a description of some of the CFD and cycle analyses performed at the NASA Glenn Research Center. As a starting point for our studies we used the thermodynamic cycle analysis, developed by Heiser and Pratt[2] under Glenn sponsorship. In that work, the authors showed that the detonation process is neither a constant volume nor a constant pressure process, but rather, a distinctive change in the thermodynamic state properties specific to the detonation process. As shown in figure 1, inlet and mechanical compression (if present) may occur in an isentropic manner as does the expansion. In their ***ideal and real (non-isentropic)*** example cases, they showed the effects of heat addition, entrance temperature and forward velocity. Their example cases used values of the ratio of specific heats and the constant pressure for air. In addition, equal values of the lower heating values of typical hydrocarbon and hydrogen fuels were used for both Brayton and PDE cycles. The calculations were performed using reference values of 400 °R. and a constant pressure specific heat of 0.24 BTU/lb-°R. The results in their paper were only meant to illustrate typical performance parameters achievable with an air-breathing pulse detonation engine (PDE). ***For a specific fuel and given flight conditions or for comparison with experimental data, the analysis needs to be applied using realistic values of the variables.*** It is further noted that their analysis was developed using the same methodology as that used for IC engines, such as the Otto and Diesel cycles. The usefulness of such cycle analyses has been amply demonstrated in the past, in spite of being a non-time dependant approach.

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In our first study,[3] we applied the analysis of reference 2 using a variety of fuel-air mixtures.

Our first task was to calculate the heat addition occurring in the PDE specific to each fuel, rather than assuming that the lower heating value of the fuel-air mixture was the correct value, as done in reference 2. In order to obtain the sensible heat release, the chemical equilibrium code, CEA, of Gordon and McBride was used.[4] In so doing, ***we found the sensible heat addition in the detonation process was 12 percent lower than in the case of deflagration and in both cases the sensible heat was less than the lower heating value of the fuel-air mixture***. This effect was attributed to the dissociation occurring during the high temperature detonation and deflagration processes. In reference 3, we chose to use standard temperature and pressure for our reference conditions, as well as for the initial conditions for the equilibrium calculations. Equivalence ratios were normally chosen to have a value of one.

As a check on our calculations, we calculated the Chapman-Jouget Mach number, using the equation, which relates the leading normal shock wave Mach number, M_{CJ}, to the non-dimensional heat addition and the ratio of specific heats:

$$M_{CJ}^2 = (\gamma+1)\frac{\tilde{q}}{\psi} + 1 + \sqrt{\left[(\gamma+1)\frac{\tilde{q}}{\psi}\right] - 1}$$

where $\tilde{q} = f\, h_{PR} / c_P T_0$. In order to obtain the same value for M_{CJ}, it was necessary to use the CEA values of the specific heat ratio and c_p of the detonated mixture. This particular approach ensures that all of the detonation characteristics are retained and are consistent.

Our second point of departure from reference 2 was to use the corresponding sensible heat release from either the detonation process for the PDE cycle <u>or</u> the deflagration process for the Brayton cycle in order to calculate the non-dimensional heat release parameter, $\tilde{q}$. This $\tilde{q}$ parameter is also used to determine the thermal efficiency of the cycles. For the PDE, the thermal efficiency is[2]

$$\eta_{th} = 1 - \frac{\frac{1}{M_{CJ}^2}\left(\frac{1+\gamma M_{CJ}^2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma}} - 1}{\tilde{q}}$$

Again, our departure from reference 2 lies in the fact that we using a lower $\tilde{q}$ value for the detonation cycle (due to greater dissociation losses) than we used for the deflagration (Brayton) cycle. Although the η_{th} of the PDE cycle still remains higher than that of the Brayton cycle (see figure 2), there is now a smaller difference in the efficiency between the two cycles. The effect of the relative decrease in the thermal efficiency has a direct impact on the propulsion parameters as will be discussed later. Our concern centered on the relative dissociation losses in a PDE and a conventional Brayton cycle process, whose combustion process occurs at significantly lower temperatures. With the background given in this section, we are now prepared to look at values of the specific thrust, impulse and fuel consumption.

<u>Cycle Results</u>

<u>Performance</u>

The impact of thermal efficiency on thrust is given by the expression[2]:

$$\frac{F}{\dot{m}_o} = \frac{1}{g_c}\left[\sqrt{V_0^2 + 2\tilde{q}\,\eta_{th} c_p T_0}\right] - V_0$$

As may be seen from the specific thrust equation, it is the product of the non-dimensional heat release and the thermal efficiency that establishes the relative thrust of various propulsion cycles. Since $\tilde{q}_{DETONATION}$ is lower than $\tilde{q}_{DEFLAGRATION}$, and the thermal efficiency difference has decreased due to dissociation effects, the thrust equation predicts that the two cycles will have equal thrust when:

$$(\tilde{q}\,\eta_{th})_{DETONATION} = (\tilde{q}\,\eta_{th})_{DEFLAGRATION}$$

If that equality is satisfied, then the two curves will have a common point and they will cross or merge with each other.

The result in figure 3, for a stoichiometric propane-air mixture, shows that the equality given above occurs at a temperature ratio of 2.25.[3] Hence, the dissociation losses have caused the PDE performance to fall below the Brayton (ramjet) cycle above that temperature ratio. Similar results occur for both the specific fuel consumption and the impulse values. The temperature ratio in figure 3 is directly related to the flight Mach number, and it is more illustrative to present results in those terms. It is important to recognize the differences in the compression that occurs with each cycle.

Typical results from reference 5 are shown in figure 4 for flight conditions at 33,000 ft altitude. The pressure and temperature at station 0 (see figure 1) were 3.8 psia and 400 °R. The heat release was calculated as described above. In these calculations, both cycles have the same ram compression, but the Brayton cycle has some additional compression due to the presence of a mechanical compressor. The PDE is designed to avoid the complexities of rotating turbomachinery. Hence, for a given flight Mach number, this results in lower temperature value, T_3, at the detonation chamber entrance than in the case of a Brayton cycle. The mechanical compression ratio, π_c, is added as an additional parameter on this figure. The bold lines represent the permissible operation range when the compressor discharge temperature is limited to 1710 °R.

Calculations were performed to examine the effect of Mach number.[5] The PDE was found to have higher specific thrust at Mach numbers up to 2.3 compared to the ramjet. It also had higher performance than a turbojet which had mechanical compression ratios below 4. This advantage of the PDE disappears as the mechanical compression of the turbojet exceeds 4. It also disappears when the ramjet flight speed exceeds a Mach number of 2.3.

The performance calculations were also examined for non-ideal process efficiencies,[5] i.e., compression, burning, and expansion. For an expansion efficiency of 0.95, the PDE performance remained lower than the Brayton at all Mach numbers. A reduction in compressor efficiency improved the performance of the PDE slightly relative to the Brayton; whereas a reduction in the burner efficiency had a larger effect on reducing the relative PDE performance. It was concluded that the PDE may be suited for a combined cycle application. Coupling of a PDE, which theoretically can provide static thrust, with a ramjet starting at Mach 2.5 to 3, followed by a scramjet at Mach 5 to 6 may be such a possibility.[5]

CFD Analysis and Results

Real Gas Effects and Recombination

In our previous work it was shown that the high temperatures associated with detonation caused a substantial amount of dissociation, yielding high concentrations of intermediate species. This process led to an 11 percent loss in the sensible heat release available for the production of thrust.[3] As shown in the last section, the consequence of this loss was to reduce the thermal efficiency and the heat release, thereby reducing the specific thrust below that of a ramjet at Mach numbers of 2.3. Since the detonation products persist in the chamber for a finite length of time, we studied the effect of recombination on heat addition.[6]

For this work, finite-rate chemistry unsteady computations were performed for an open-ended tube filled with hydrogen-air mixtures at standard conditions. It was found that some recombination occurred during the residence time in the tube. This resulted in a smaller amount of sensible heat loss than shown in the previous portion of this paper. The differences are shown in figure 5. The main conclusion from our previous study[6] is that the dissociation loss for an equivalence ratio of 1.0 reduces the sensible heat release by 16.7 percent relative to the heating value, when chemical equilibrium was assumed. Recombination in the PDE reduces this loss to 10.8 percent.

For an equivalence ratio of 0.6, dissociation effects reduced the sensible heat release by 4.7 percent when equilibrium was assumed, and recombination reduces the loss to 0.6 percent. The study also showed that the fuel specific impulse at 0.6 was higher than that at an equivalence ratio of 1.0, although the thrust at 0.6 was lower.

Comparison of CFD and Cycle Analyses with Experimental Data

In our latest work, the analyses described above were compared to the hydrogen-air experimental data obtained at the Wright Labs.[7] In figure 6, our data are shown on the impulse-equivalence ratio plot of reference 8. The open squares represent our thermo cycle upper performance limit and the open diamond symbols represent our finite rate CFD results. The CFD results are in excellent agreement with the data and the cycle results show the potential improvement, amounting to 200 to 300 seconds of impulse.

This work was carried out using hydrogen-air mixtures due to the complexities of finite rate chemistry modeling for hydrocarbons as well as the lengthy computational time required for unsteady CFD. A reduced hydrocarbon scheme is under study, which should allow us to perform similar computations.

Propulsion Performance Variation with Mach Number

Based on the results obtained with the hydrogen-air calculations, it was decided to estimate the recombination effect that would occur in the propane-air detonation used in our previous work and shown in figure 4 Since the dissociation losses reduced the sensible heat release by 16.7 percent relative to the heating value when chemical equilibrium was assumed and the recombination reduced the losses to 10.8 percent. A decrement of 5 percent was assumed for the sensible heat loss relative to the lower heating value of the propane fuel. The non-dimensional $\tilde{q}$ described previously was calculated using a sensible heat release that was 5 percent lower than was used in figure 4. The new thrust results are shown in figure 7 as a function of Mach number. It may be seen that the larger $\tilde{q}$ has increased the relative performance of the PDE relative to the Brayton cycle. The PDE specific thrust is now comparable to the ramjet performance out to a Mach number of approximately 3.0. However, this result may be somewhat optimistic in so far as the effect of recombination has not been taken into account for the Brayton cycle.

The specific impulse is also shown on the right margin of the plot. The corresponding fuel consumption variation with Mach number is shown in figure 8. It is observed that the PDE's fuel consumption is not significantly different than that of a turbojet with a compression ratio of 10.

Concluding Remarks

Some issues related to the use of PDE systems require improved physical understanding. The three noted here are related to combined system applications, to exhaust systems and to the low ratio of thrust/frontal area.

Combined Cycle System

The PDE could be considered for applications similar to those of rocket- boosted systems, although this does not utilize the inherent static thrust capability of a PDE.

Since a PDE has the ability to provide thrust at static conditions, the possibility of a combined- cycle propulsion system capable of flight from ground to the high Mach number required for scramjets ($\approx$Mach 5.5) is conceivable. However, sufficient thrust must be generated in order to propel the weight of the entire vehicle. The additional thrust may be achieved through the use of a backpressure device or nozzle.

Nozzles and Exhaust Systems

The CFD results in this paper have focused on open-ended detonation tube performance. The thrust surface for such a device is limited to the closed end of the tube. Some form of a nozzle would be expected to provide additional surface area for thrust production. Some results have shown that the thrust of a PDE may be increased substantially by the use of a backpressuring nozzle.[1,9,10] These increases may amount to factors approaching 50 to 100 percent of the impulse. Values of this magnitude would establish the PDE as a strong contender for combined cycle systems.

It is noted that our cycle analysis, unlike the CFD, assumed an isentropic processes without any specification of the exhaust system. The performance results showed a gain of 200 to 300 seconds of impulse; which is hardly equivalent to the nozzle gains mentioned above. Further work is required to clarify this issue. In particular, the question that needs answering is whether or not the cycle analysis represents the upper performance limit for a PDE. Isolation of the exhaust blast wave from adjacent tubes may be required. This problem could be addressed with separate exhaust nozzles where the individual tubes are centered about a center body or a sculpted plug nozzle with dividing struts between each exhaust stream. Other aerodynamic concepts involving fluidic devices are possible such as vortex flow control.[11]

Thrust per Unit Frontal Area

In spite of the promise of significantly higher performance, the PDE suffers from a low specific thrust to frontal area or mass flow ratio. The possibility of using multiple chambers has been considered by a number of investigators. These schemes employ either a stationary or concentric rotating bank of PDE tubes, which are fired in pairs on a sequential basis. The effect of these multiple tubes firing at sufficiently high frequency may assist in overcoming the low thru-flow issue.

Finally, it is noted that the PDE may have practical engineering advantages over a gas turbine engine, i.e., simplicity, fewer moving parts, lighter weight and lower cost. In order to realize these benefits, the effects of high temperature and high internal flow velocity on heat transfer and viscous losses as well as fatigue and leakage issues related to multi-cycle operation and valving require attention. These items have not been addressed in this paper and must await practical demonstration.

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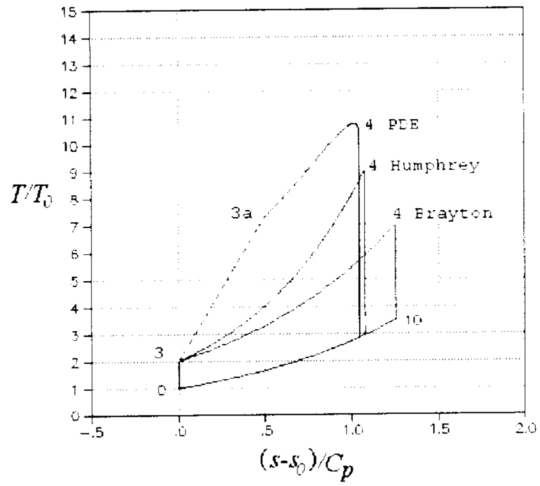

Figure 1. Ideal Cycle Analysis for the Brayton, Humphrey, and PDE Processes.[2]

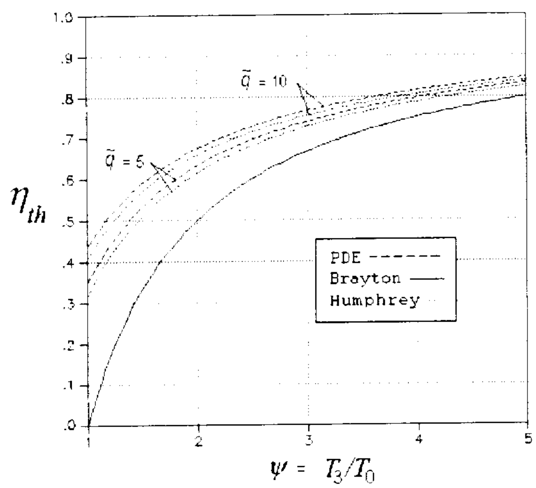

Figure 2. Thermal efficiency of the Brayton and PDE Cycles for equal values of $\tilde{q}$.[2]

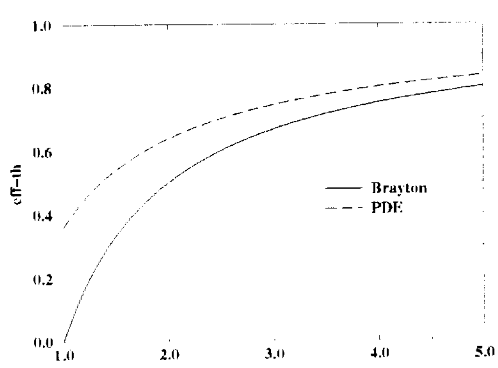

Figure 3a. Thermal efficiency for unequal values of $\tilde{q}$, stoichiometric propane-air.[3]

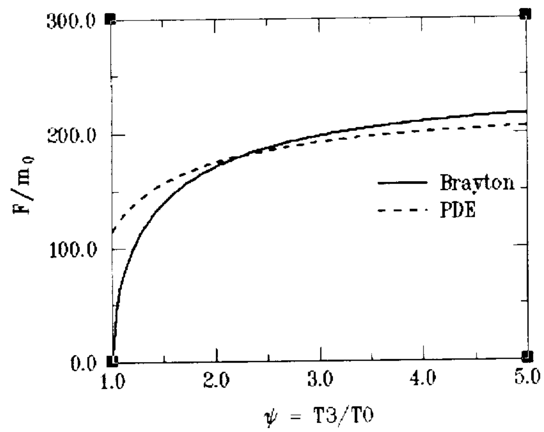

Figure 3b. Specific thrust for the PDE and Brayton Cycles versus temperature ratio, stoichiometric propane-air.[3] (units same as in figure 4)

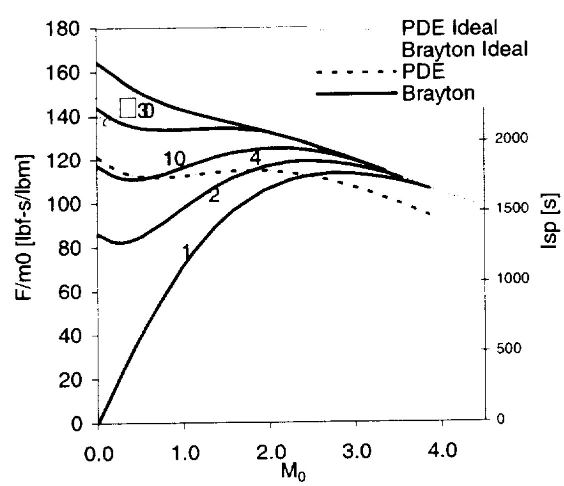

Figure 4. Specific thrust versus Mach number for Brayton and PDE cycles, stoichiometric propane-air.[5]

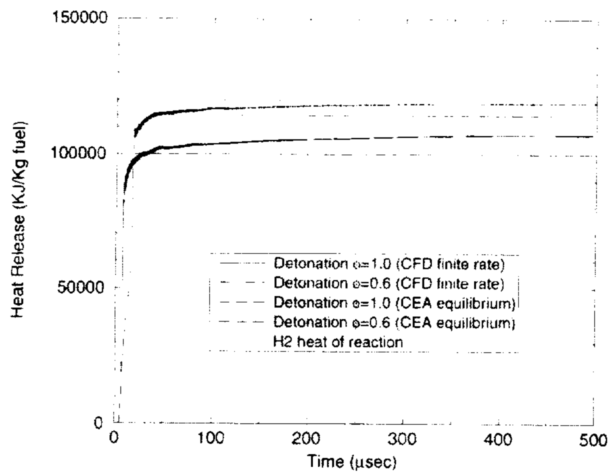

Figure 5. Sensible heat release for hydrogen-air, equilibrium, and finite rate reaction (lower curve is stoichiometric).[6]

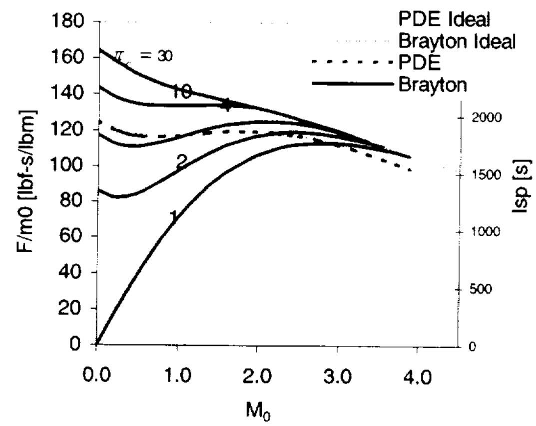

Figure 7a. Specific thrust versus Mach number including recombination effect, stoichiometric propane-air.

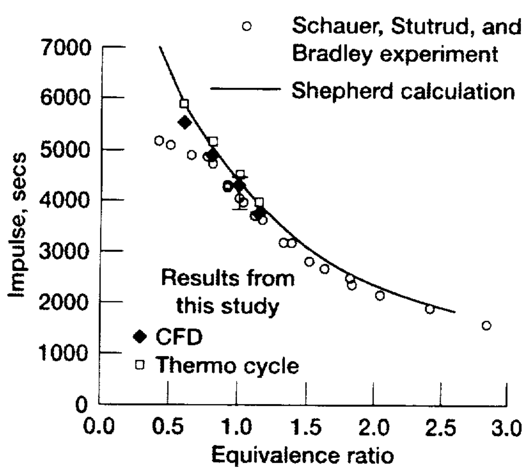

Figure 6. Comparison of analyses with WL data, hydrogen-air.[7]

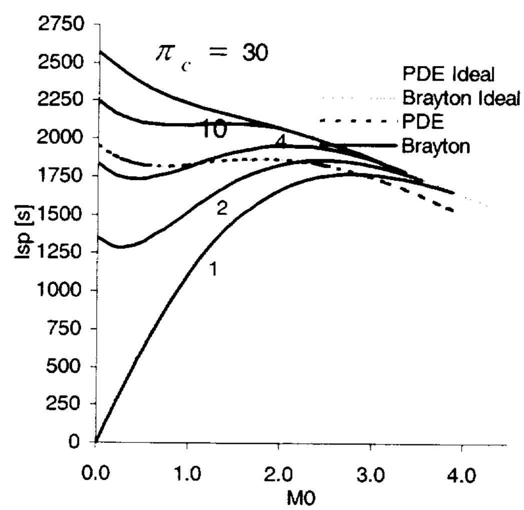

Figure 7b. Specific fuel consumption and Mach number variation, including recombination effect, stoichiometric propane-air.

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13. ABSTRACT *(Maximum 200 words)*

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